

## Nozomi Earth Swingby Orbit Determination

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The Nozomi mission has been disrupted by several anomalies, including a maneuver under burn, loss of the low gain S-band transponder and a large solar flare which damaged many spacecraft subsystems. These anomalies placed constraints on the Nozomi operation that resulted in severe degradation of navigation data collected during critical mission phases. Resultant large navigation errors could potentially have resulted in the loss of scientific return from the mission. This paper describes the Jet Propulsion Laboratory effort to navigate Nozomi, in conjunction with Japan's Institute of Space and Astronautical Science, past two Earth swingbys using differential tracking techniques to successfully place the spacecraft on a trajectory reaching Mars in December 2003.

### INTRODUCTION

Japan's Institute of Space and Astronautical Science (ISAS) launched Nozomi, its nations first mission to the planet Mars in 1998. Nozomi is a cooperative mission between ISAS and the National Aeronautics and Space Administration (NASA). The NASA contribution includes navigation and tracking services provided by the Jet Propulsion Laboratory (JPL). Both ISAS and the Navigation and Flight Mechanics Section at JPL perform orbit determination for Nozomi in parallel. In this paper, information regarding the JPL orbit determination effort supporting the transfer trajectory to Mars, which was conducted under extremely difficult circumstance, is presented. The mission profile, spacecraft and operations are characterized, followed by a discussion of the anomalies that adversely affected the mission and the quality of navigation data. The orbit determination process, including models, estimation procedure and data considerations are discussed. Special emphasis is placed on unique tools available at JPL that helped significantly reduce navigation uncertainty for critical events. Orbit determination results are presented for the two Earth swingbys.

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## Overview and Nominal Mission Design

The Nozomi spacecraft, its launch and mission design were conceived and executed by ISAS. The scientific objectives of the mission were to study the structure and dynamics of the Martian upper atmosphere and its interaction with the solar wind.

Nozomi, whose name means “hope” in Japanese, was known as Planet-B prior to launch. It was launched on July 3, 1998 using the newly developed M-V launch vehicle. The spacecraft was initially placed into a highly elliptical phasing orbit. An outbound flyby of the moon on September 20, 1998 increased apogee to the vicinity of the weak stability boundary (approximately 1.7 million km), where solar gravitational perturbations increased the spacecraft velocity relative to the Earth. Nozomi then performed an inbound lunar swingby to gain additional speed followed only two days later by a powered Earth swingby on December 20, 1998. The large Trans Mars Insertion (TMI) burn at the final Earth perigee was intended to place the spacecraft into a heliocentric transfer trajectory. The nominal mission plan called for Mars orbit insertion on October 10, 1999 (Ref. 1). This would place the spacecraft in a 150 km by 50,000 km orbit allowing observations of the Martian surface, its lower atmosphere and ionosphere, magnetosphere and intrinsic magnetic fields at periapsis. Other experiments include solar wind measurements, imaging of the moons Phobos and Deimos and determination of the vertical structure of the Martian atmosphere by radio occultation observations. The expected mission lifetime in Mars orbit is about two years.

## Spacecraft Description

Nozomi is very small by interplanetary standards. It had an injected mass of 540 kg, including 280 kg of propellant. The spacecraft bus measures only 1.6 m by 0.6 m and supports two fixed solar panels, which span 6.2 m from tip-to-tip. The spacecraft is spin stabilized, with attitude control maintained by ten mono-propellant 2.3 N thrusters. Large maneuvers are performed by a single bi-propellant 500 N engine.

Communications is via two low gain antennae (LGA-A and LGA-B) which provide uplink and downlink at S-band frequencies, and a high gain antenna (HGA) which supports S-band uplink and both S-band and X-band downlink. LGA-A is located on the spacecraft spin axis, mounted on top of the antenna feed of the 1.6 m parabolic HGA mesh dish. LGA-B is mounted on the opposite side of Nozomi and is offset 1 m from the spin axis.

The spacecraft also functions as a technology demonstration mission. Innovations include a lightweight composite mesh high gain antenna, surface-mounted electronics circuits, high-efficiency solar cells (16.8%), Nickel-Metal-Hydride battery and composite He tank.

The scientific payload has a mass of 36 kg. Instruments include a visible light camera, ultraviolet spectrometer, neutral and ion mass spectrometers, dust counter and a suite of fields and particle sensors. NASA provided the Neutral Mass Spectrometer for the science payload that was built by the Goddard Space Flight Center with Dr. H. Neumann as Principal Investigator.

## Cooperative Operations

The primary tracking station to support Nozomi is the 64 m antenna at the Usuda Deep Space Center, Japan. This station provides telemetry, command and navigation data collection functions. One limitation of using only domestic Japanese tracking is that ISAS would only be in contact with the spacecraft about 10 hours per day. One of NASA's contributions was to provide supplemental tracking for navigation data while Nozomi was out of view of the Usuda station. This support is provided by JPL's Deep Space Network (DSN) tracking stations at Goldstone and Madrid. Close coordination between ISAS and JPL were required to integrate the scheduling of passes from the two networks. Note that the DSN stations do not provide telemetry and command functions.

Navigation and mission design for Nozomi is performed by ISAS. But as part of the DSN support, orbit determination is conducted in parallel with ISAS by JPL, with each group generating solutions based on data collected from their respective tracking networks. This scenario provides several distinct advantages to the project. First, the JPL solutions contain data from widely spaced DSN tracking stations that increases the navigation information content, especially for short data arcs. Second, spacecraft events, such as command uplinks and maneuvers, were scheduled during passes at the Usuda tracking station in Japan. As a result, maneuver design and initial assessment was often derived from JPL solutions based on DSN tracking data explicitly scheduled to bracket these critical activities. Lastly, the fact that ISAS and JPL used different models, data sets and calibrations, allowed an independent verification of orbit determination solutions. Even though ISAS did not levy explicit navigation accuracy requirements on JPL, the favorable comparison of orbit determination estimates greatly increased confidence in the navigation process.

## Mission Anomalies

The first six months of the mission proceeded nominally. The final event of the cis-lunar phase was the 420 m/s Trans Mars Insertion burn (TMI), which placed the spacecraft into heliocentric orbit. This maneuver was critical because each m/s of error in TMI execution performance would cost 3.5 m/s during a TMI correction burn (TMIC) about 12 hours after periapsis (Ref. 2). TMI occurred out of contact with ground stations and the first post maneuver tracking pass was over the DSN Goldstone complex. The JPL navigation team quickly determined that a 100 m/s under burn had occurred. ISAS immediately began preparations for the correction maneuver, which has to be performed during the next pass at the 64 m domestic tracking facility at Usuda, Japan. ISAS determined that the under burn was caused by a partially opened oxidizer valve, which they were able to clear in time for the 340 m/s TMIC.

The large TMIC maneuver left the spacecraft with insufficient propellant to enter Mars orbit. A new plan was developed which took advantage of the low  $C_3$  Earth-Mars transfer trajectory occurring in the spring of 2003. This plan added three full solar orbits, followed by an Earth swingby in December 2002 and a second Earth swingby in June 2003. Insertion into Mars orbit was scheduled to occur in January 2004 with barely sufficient propellant to achieve all

mission science objectives. The one significant drawback to the redesign was the long delay in arrival at Mars – well beyond the nominal design life of the mission.

The second major anomaly was the loss of the S-band transponder in early July 1999. This anomaly disabled downlink via the low gain antenna systems as well as the ultra-stable oscillator, needed for the atmospheric occultation experiments at Mars. Following this event, communications with the spacecraft could only be conducted by using the low gain S-band antenna for uplink and the high gain X-band antenna for downlink.

The final blow occurred on April 26, 2002 when Nozomi was near aphelion. A major solar proton event caused the loss of contact with the spacecraft. ISAS determined that significant damage had been done. A short had occurred on the main spacecraft bus. Subsystems affected included the main engine (needed of insertion into Mars orbit) and telemetry. Propellant line heaters were also disabled, which caused hydrazine to freeze because of the large distance to the sun.

## NAVIGATION

ISAS attempted to repair the short circuit through out the summer of 2002. One consequence of this operation was that it frequently disabled the downlink. They also developed procedures that worked around the shorted-out subsystems and allowed limited operation of many spacecraft functions. Controllers turned on science instruments to warm the spacecraft. This, in combination with the reduction in distance from the sun, resulted in the fuel lines thawing by the end of August 2002. At this point, ISAS was forced to postpone work on the short circuit and restore the downlink. The next ten months would be devoted to placing the spacecraft on a trajectory that would perform the two upcoming Earth swingbys with sufficient accuracy to reach Mars.

### Mission Challenges

A major challenge during this phase of the mission involved a tradeoff between the competing requirements of power, thermal constraints and telecommunications and navigation. Until this time the spacecraft had been outside the Earth's orbit. It was possible to point the fixed high gain antenna at the Earth and have sufficient sunlight on the fixed solar panels to power the spacecraft. The high gain antenna could not be pointed at the Earth for the six-month period starting just prior to the first Earth swingby to just after the second swingby because the Earth-Probe-Sun angle was 90°. ISAS devised a plan to employ a broad side lobe in the high gain antenna pattern that was approximately 100° off bore sight. In this configuration there was sufficient link margin to acquire degraded range data only to a distance of 6,000,000 km and telemetry to a distance of 8,000,000 km. This meant that there would be no downlink from the spacecraft for the central 4 months of the Earth-Earth transfer. The link study also indicated that it was only marginally possible to acquire Doppler data for the entire Earth-Earth transfer.

The primary concern for navigation was the  $\Delta V$  cost to correct errors in the trajectory for the swingbys. A 10 km position error in the first swingby would require a 21 m/sec correction maneuver performed 21 days prior to the second swingby. If the maneuver were delayed one

week it would require a 55 m/s burn. So the navigation challenge was two fold: deliver the spacecraft to the first swingby as accurately as possible and then reestablish trajectory knowledge as quickly as possible after the resumption of tracking to permit an early correction maneuver leading into the second swingby.

The task of reestablishing the orbit was especially challenging because of the short turn around time required and the fact that the tracking data would be degraded (or possibly non-existent). ISAS proposed that we employ the DSN interferometric tracking capabilities to augment traditional Doppler and range data types to improve navigation accuracy for the anticipated short data arcs.

### DDOR Overview

Delta Differential One-way Ranging (DDOR) is a radiometric tracking technique used to obtain instantaneous plane-of-sky position of deep-space spacecraft (Ref. 3). The technique is an extension of Very Long Baseline Interferometry (VLBI). In DDOR, the signal phase of two or more signals from the spacecraft of interest are simultaneously recorded at two tracking stations and then differenced first between stations and then between tones. This provides an uncalibrated measurement of the bearing to the spacecraft with respect to the baseline between the two stations. The measurement is calibrated by recording the noise from a natural radio source (quasar) that is nearby in the plane of the sky at proximate time and frequencies close to the spacecraft tones.

The DDOR measurements for Nozomi utilized the VLBI Science Receivers (VSR) placed at the three Deep Space Network (DSN) tracking stations. The receivers are based on an earlier Radio Science Receiver (Ref. 4) with modifications to improve utility in DDOR and VLBI measurements. An RF input signal from a DSN antenna is initially mixed down to near 300 MHz, and a 100 MHz band near 300 MHz is digitally sampled. Subsequent digital signal processing within the receiver can be configured to simultaneously record several narrower channels within the 100 MHz band. During a DDOR tracking pass, channels are configured to capture each of several spacecraft tones. Quasar noise recordings are made of wider channels at frequencies centered on each spacecraft-recording channel.

Subsequent processing on both the spacecraft and the quasar recording is performed with software. The signal phase of each spacecraft tone is determined by correlating the measured samples with a local model. Typically, the local model is developed by applying a phase-locked loop to the samples of the strongest spacecraft signal, although an externally developed model can also be used. Once the phase for each tone at each station is determined, they are differenced between stations and tone pairs to produce the difference of arrival time of the spacecraft signal between the two stations. The quasar recording, containing radio noise from the source, is processed by cross-correlating the recordings at the two stations for each recorded band to determine the time difference for the quasar signal arrival. Since the quasar plane-of-sky position is well known, the measured time differences for the quasar are used to calibrate the effects of station timing

error, receiver frequency response, and most of the delay difference from differing media in the spacecraft measurement.

This DDOR system, including the VSR and subsequent processing software, has been used to make measurements for several spacecraft including Nozomi as well as Mars Odyssey, Mars Global Surveyor, MER-a, MER-b and Mars Express. The measurement precision of DDOR scales inversely with the frequency span between the tones and the separation of the recording stations. For widely spaced tones ( $> 20$  MHz), other error sources contribute (see Ref. 3). Measurement precision as low as 2 nrad have been achieved with this system on spacecraft equipped with radios producing widely separated tone. For Nozomi, a tone separation of 2 MHz was typically available producing DDOR observables with approximately 50 nrad precision.

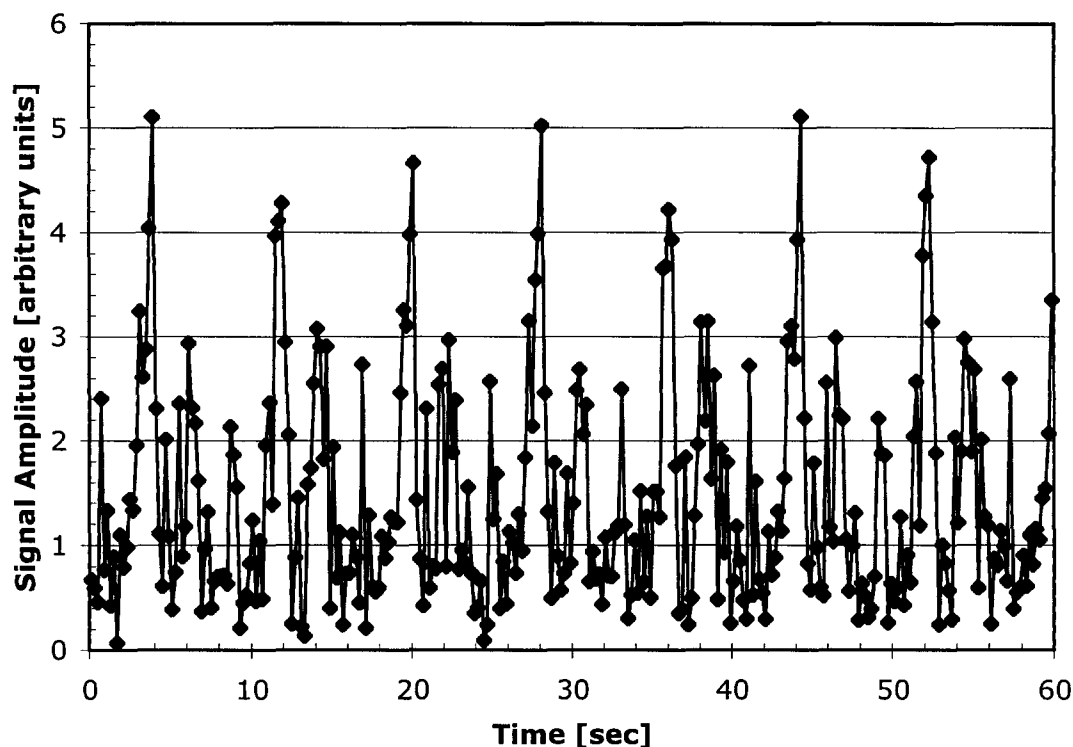
### DDOR Processing

Since the Nozomi radio system was not equipped to generate tones with the 20 to 40 MHz spacing commonly used for DDOR, the ranging system was adapted for use in DDOR. A nearly 1 MHz range tone coherent with the carrier signal was uplinked to the spacecraft and then returned by a transponder. The harmonics of the range tone up to the sixth harmonic were visible in the received spectrum during tests prior to the first DDOR measurement with Nozomi. The higher harmonics were weak and could only be observed during the phase of the mission when the spacecraft's high gain antenna was on Earth point. Consequently, a tone separation of about 2 MHz ( $\pm 1$  MHz) was available for most of the DDOR measurements.

During several of the DDOR measurements, Nozomi's antenna was purposely off Earth point resulting in difficulty in making a continuous measurement of signal phase. The carrier signal  $P_c/N_0$  varied from 15 to 25 dB-Hz during these measurements and had instability induced by the spacecraft's line-of-sight to Earth passing through variations in the antenna's gain pattern as the spacecraft spun. Drastic variations in signal amplitude were seen (Figure 1), and, during the low points in signal amplitude, the phase could not be measured. Upon differencing the series of phase measured at two stations, it was possible to connect the difference phase across the breaks, due to the detailed modeling of the station positions and Earth motion used in generating the DDOR observable.

DDOR acquisition typically requires the coordination of a number of activities. Candidate quasars that have strong signals and are observable near the spacecraft in the overlap between intercontinental tracking complexes must be identified. Then a long term tracking request is submitted, negotiated and integrated into a schedule that is free of conflicts with other users of the network. A sequence team then generates and verifies commands that are uploaded and executed to enable the DDOR tone hardware on the spacecraft. The sequence team also generates "keyword" file that specifies the number, duration and sequence of observations taken during the pass. The "keyword" files are also used to control the DDOR recording equipment and for pointing of the tracking stations involved. Separate DDOR frequency predicts and ionosphere calibrations are also needed to support this activity. DSN tracking of Nozomi does not require interaction

with the ISAS sequence team, so the JPL navigation team constructed pseudo-“keyword” files for use in the DDOR acquisition.



**Figure 1 Nozomi Sample Carrier Signal on Jan 4, 2003 DDOR Pass**

### Orbit Determination

Orbit determination was performed using a pseudo-epoch state least-squares filter that determines the spacecraft trajectory by estimating the spacecraft’s epoch state and various parameters that model the dynamical environment that influences the spacecraft motion. These dynamical influences include trajectory correction maneuvers, reorientations and solar pressure. Stochastic non-gravitational accelerations are used to account for small errors in the solar pressure model due to the uncertainties in the spacecraft attitude. Stochastic, per pass range biases are employed to account for range calibration errors. The contributions of the following errors were considered in the covariance: ionosphere, troposphere, station locations, Earth ephemerides, polar motion, UT1 and quasar location. A Doppler bias was estimated to remove the effect of spin-induced Doppler bias.

The orbit determination filter configuration is presented in Tables 1, 2 and 3 and the data weights employed are shown in Table 4.

**Table 1**  
**DETERMINISTIC ESTIMATED PARAMETERS**

<b>Parameter</b>	<b>A Priori <math>1\sigma</math> Uncertainty</b>	<b>Comments</b>
Spacecraft State	Position: 1000 km Velocity: 100 m/s	
Deterministic Maneuvers	100% of each component	
Reorientation Maneuvers	1 mm/sec	Z component only
Solar Radiation Pressure Scale Factor	20% of nominal	
Spin Induced Doppler Bias	1 Hz	

**Table 2**  
**STOCHASTIC ESTIMATED PARAMETERS**

<b>Parameter</b>	<b>A Priori <math>1\sigma</math> Uncertainty</b>	<b>Comments</b>
Non-Gravitational Acceleration	Radial: $1.0D-11 \text{ km/s}^2$ X & Y: $1.0D-12 \text{ km/s}^2$	
Per Pass Range Bias	10 m	

**Table 3**  
**CONSIDER PARAMETERS**

<b>Parameter</b>	<b>A Priori <math>1\sigma</math> Uncertainty</b>	<b>Comments</b>
Station Locations	Formal DSN station location covariance	From IM DSN Station Location Update and Plans REVISED, Folkner et al., 16 April 2003.
Troposphere	1 cm dry 4 cm wet	Zenith range delay
Ionosphere	5 cm day 1 cm night	Range delays, X band
UT1	$3.2D-4$ seconds	
Pole Location	$7.5D-9$ radians	X and Y components
Earth/Mars Ephemeris	Formal DE403 ephemeris covariance	From IM JPL Planetary and Lunar Ephemerides, Standish, 22 May 1995.
Quasar Position	$2.788D-7$ degrees	1 milli-arcsecond

**Table 4**  
**DATA WEIGHTS**

<b>Data Type</b>	<b>Uncertainty</b>	<b>Comments</b>
Coherent Two-Way Doppler	1.0 mm/sec	60 second count interval
Ranging data	10 meters	
DDOR	1.0-3.0 nsec	Depending on measurement quality

## **RESULTS**

DSN support for Nozomi resumed on October 2, 2002 following partial recovery from the solar flare event. A tracking pass at the DSN Goldstone complex was typically allocated immediately prior to the Japanese Usuda tracking pass in which a maneuver was scheduled to take place. JPL would rapidly produce an orbit solution and forward it to ISAS about 6 hours before the maneuver execution. ISAS would then verify the consistency of the ISAS and JPL solutions, design, uplink and perform the maneuver near the middle of the Usuda pass.

### Earth Swingby 1

JPL based its first post recovery solution on just three passes of DSN tracking. Data acquired during this phase of the mission were still of good quality. ISAS executed DV14 in three parts on October 5 and 6, 2002. This 17.6 m/s maneuver moved the Earth closest approach distance from 76,000 km to 28,000 and shifted the time of periapsis earlier by 30 minutes (see Figure 2).

Following DV14, ISAS began turning Nozomi several degrees off Earth point to improve the spacecraft thermal and power state. Consequently, noise in the Doppler and range tracking data gradually increased by approximately 100% over the data arc. Data weights in the estimation filter were adjusted accordingly. Data noise in the DDOR observations were less effected by this situation. The JPL solution included 20 passes of Doppler and range during a 49-day data arc. The DSN also acquired 9 DDOR measurements (four on the North-South baseline and five on the East-West baseline). These data were used in support of DV15, performed in two parts on November 22, 2002. This 3.3 m/s maneuver refined the trajectory in the B-Plane and adjusted the arrival time earlier by an additional 30 minutes (Figure 3).

The final trim was DV15c, a 1.3 m/s maneuver executed on December 16, 2002 (see Figure 3). Tracking data quality supporting this burn continued to degrade as the

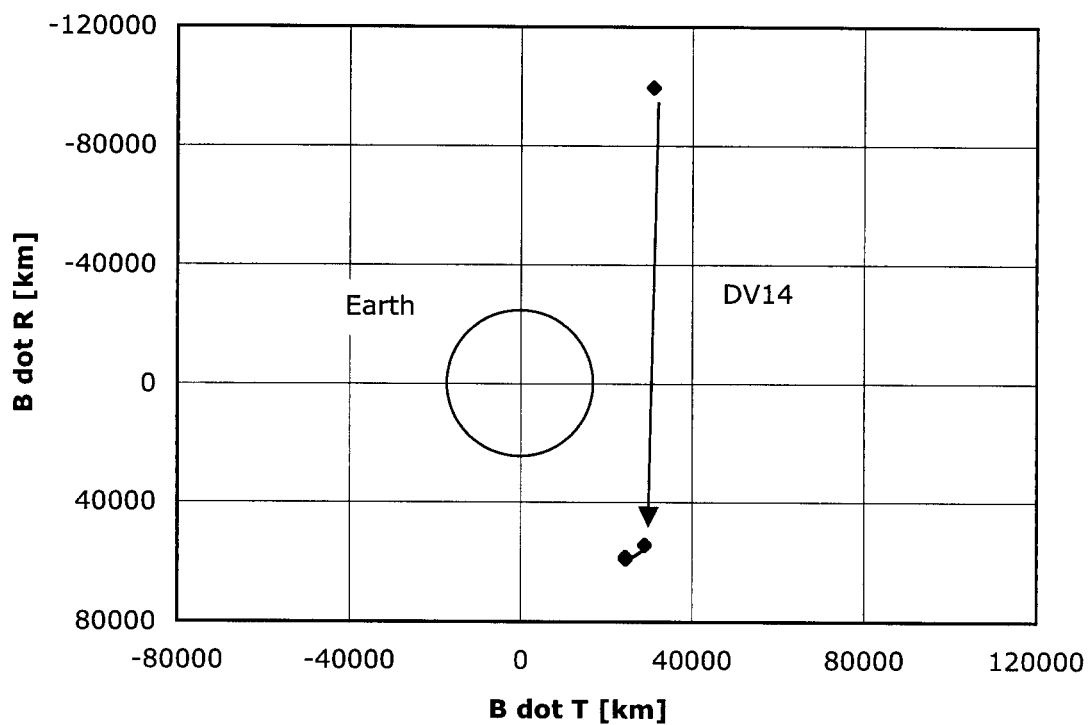
spacecraft orientation was adjusted a little farther off Earth point to  $5^\circ$  (see Figures 4a, 4b and 4c). The JPL solution for the maneuver was a 70-day data arc including 36 passes of DSN Doppler and range data plus 11 DDOR measurements (six on the North-South baseline and five on the East-West baseline).

Following DV15c Nozomi was turned approximately  $90^\circ$  off Earth point it was no longer possible to point the spacecraft in the vicinity of the Earth and maintain power. Nozomi was turned approximately  $90^\circ$  off Earth point, which improved the power and thermal situation. It was also possible that telemetry downlink and navigation data could be collected on a strong side lobe in the high gain antenna pattern. The orientation was stepped to  $85^\circ$ ,  $97^\circ$  and  $105^\circ$  off Earth point on December 17, 18 and 19, 2002 respectively, to assess capabilities at different attitudes. Doppler and range noise both increased significantly. The ground receiver would drop in and out of lock about 10 times per second due to strong asymmetries in the high gain antenna pattern and multipath effects. The range data integration time was increased to counteract these factors, but it was found range data could not be acquired at the  $105^\circ$  orientation. It was also determined that the DSN could not acquire valid Doppler data in the off-point orientations. The Doppler exhibited discrete families of biases data which could not be distinguished from the spin bias (Figure 5). The quality of the DDOR measurements were less effected because the differencing removed the common data noise, but the processing technique had to be modified to deal with the short lockup periods. Commanding was less affected by the Earth off-point situation because it employed the two low gain antennas. The uplink was changed from right hand circularly polarized to left hand circularly polarized to improve signal margin. The spacecraft was left in the  $105^\circ$  attitude so its orientation would drift back into the  $95^\circ$  range following the swingby.

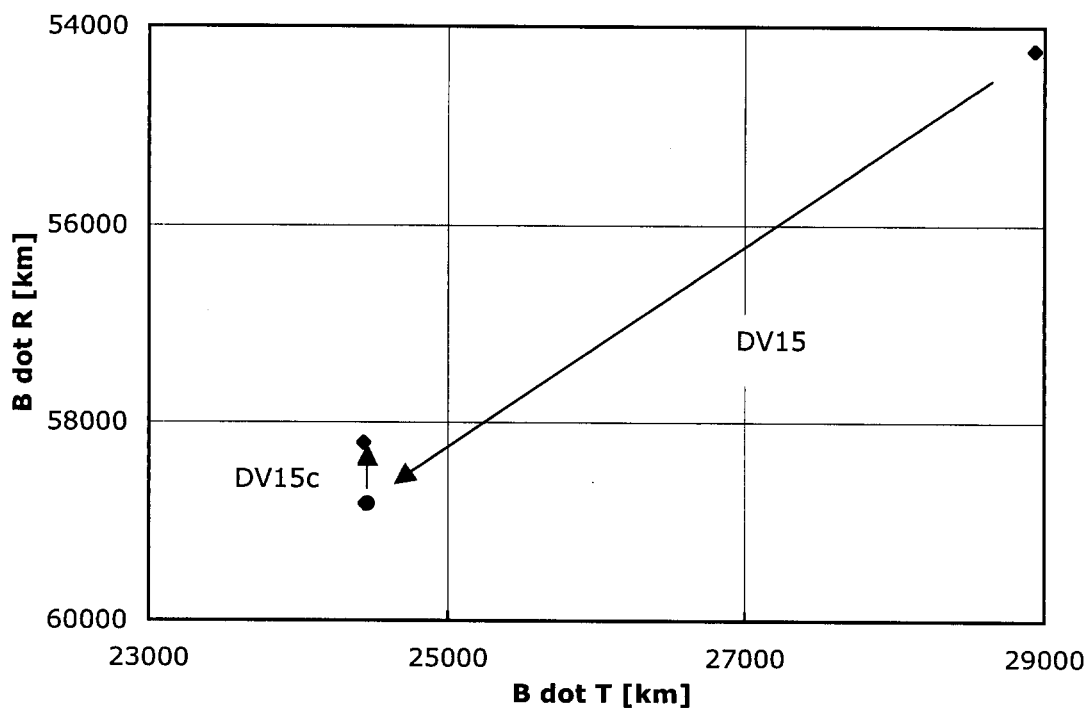
Earth swingby 1 occurred at December 21, 2002. This event was not observed because the spacecraft orientation precluded the receipt of navigation data for four days centered on periapsis. Orbit determination using degraded data from before and after the swingby found that closest approach occurred on December 21, 2002 07:36:59 UTC at an altitude of 29509.5 km. The B-vector error was 24.7 km.

The DV15cc maneuver was conducted on January 4, 2003 to correct for the inbound delivery error. This 3.1 m/s maneuver was conducted in two parts. The spacecraft declination increased to  $65^\circ$  north following Earth swingby 1. So DSN tracking could only be conducted from its northern hemisphere complexes at Goldstone and Madrid. The JPL solution supporting this event was a post swingby 10-day data arc using six passes of badly degraded range data. Another five days of range data were acquired including two DDOR measurements on the East-West baseline to verify the outgoing asymptote. Nozomi was turned to  $57^\circ$  off Earth point on January 28, 2003. This orientation was optimized for power and thermal considerations because there was no longer sufficient signal margin to support downlink as the spacecraft receded from the Earth.

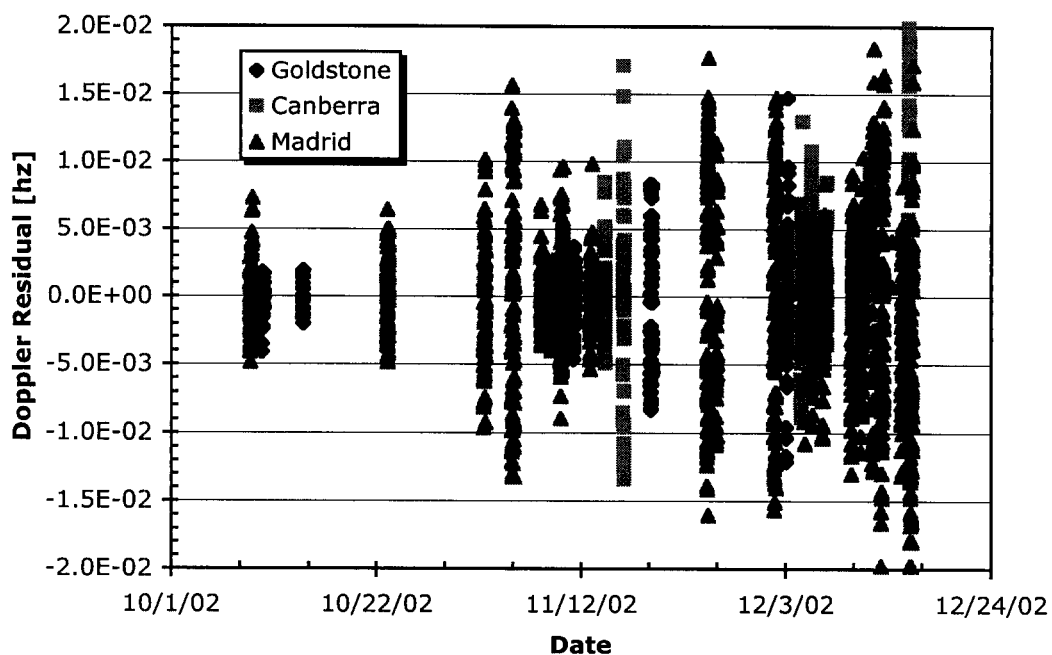
A list of Earth Swingby 1 phase events is presented as a list in Appendix 1 and in graphical format in Appendix 2.



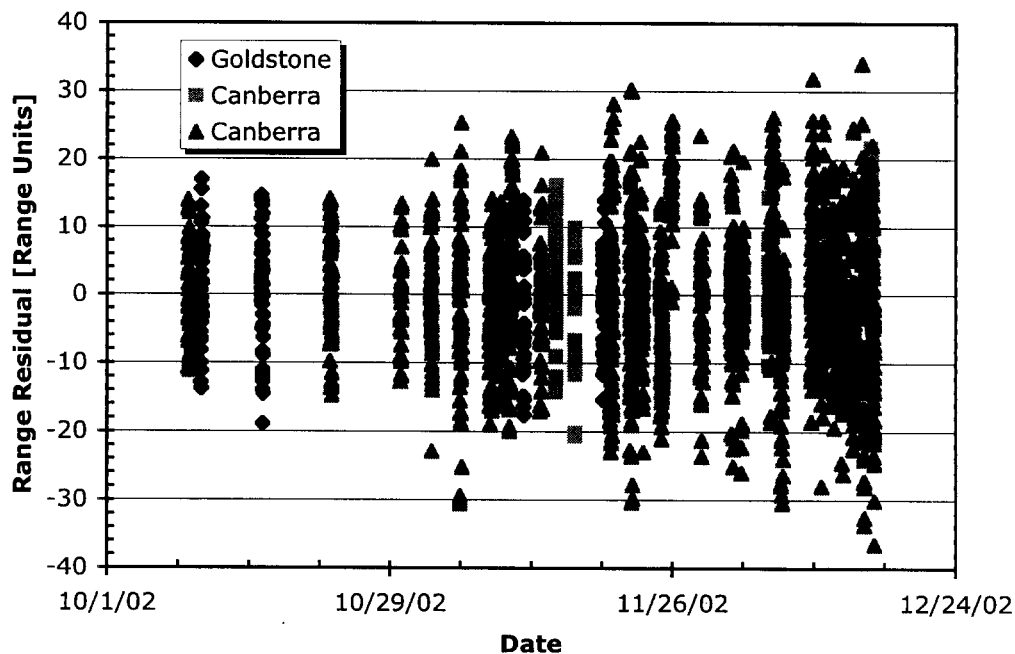
**Figure 2 Earth Swingby 1 B-Plane**



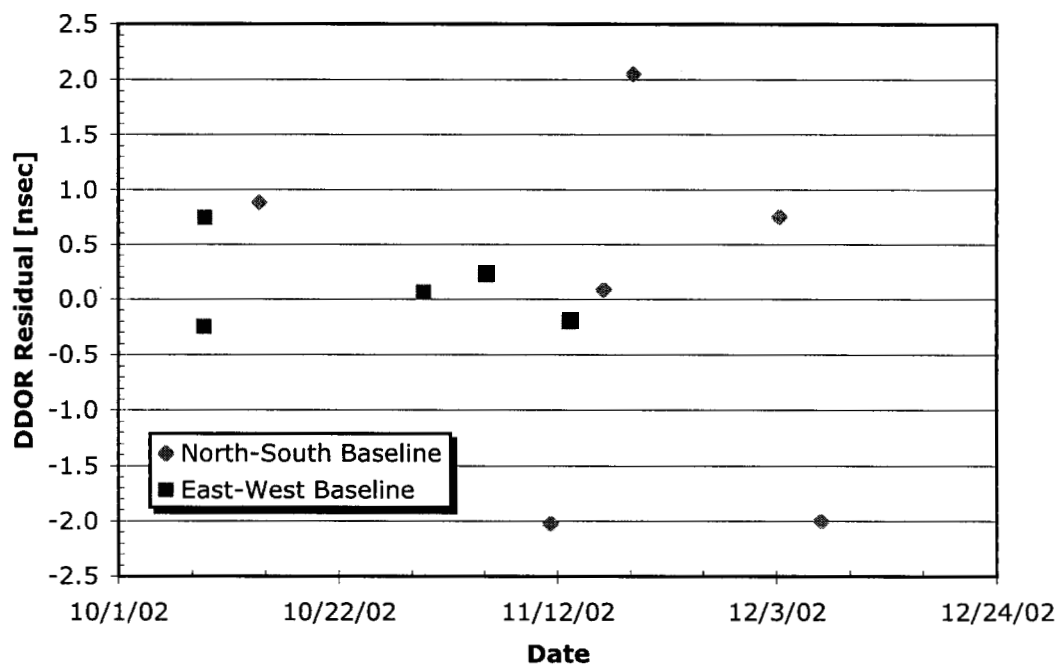
**Figure 3 Earth Swingby 1 Final B-Plane**



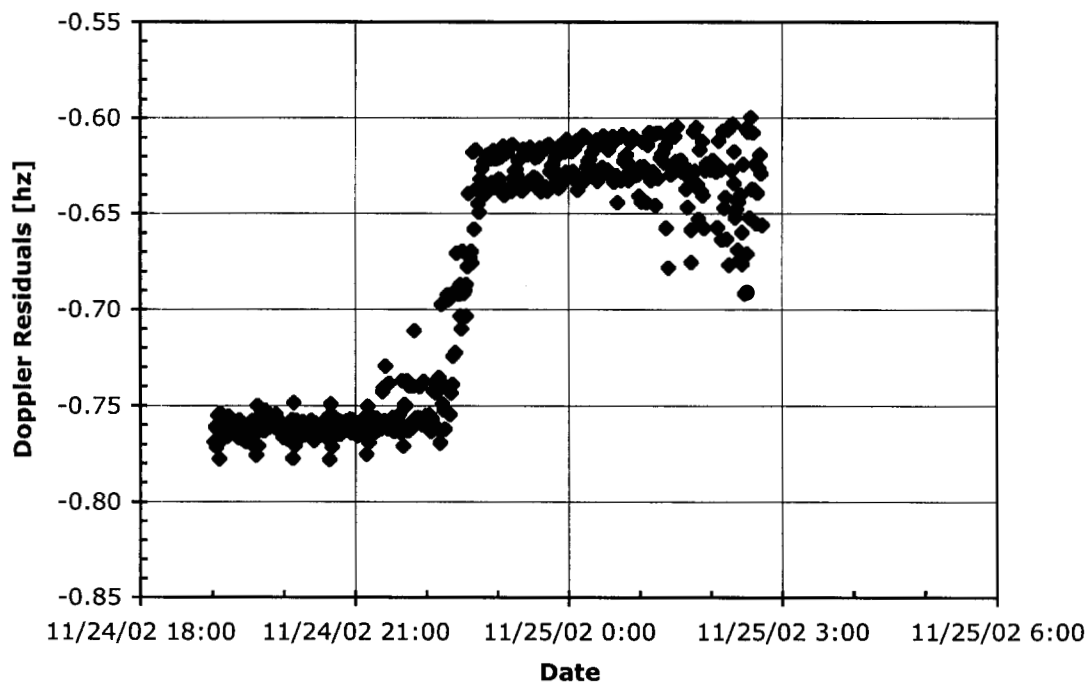
**Figure 4a Earth Swingby 1 Doppler Residuals**



**Figure 4b Earth Swingby 1 Range Residuals**



**Figure 4c Earth Swingby 1 DDOR Residuals**



**Figure 5 97° Off Earth-Point Doppler Residuals**

## Earth Swingby 2

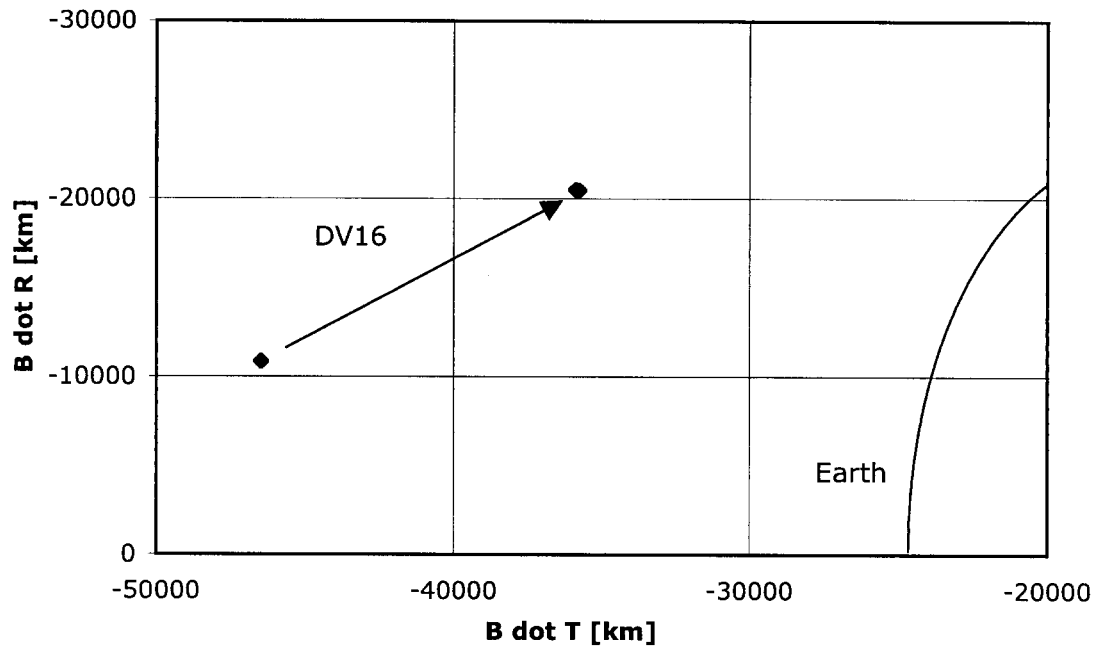
ISAS commanded Nozomi into the  $95^\circ$  off Earth point attitude on May 19, 2003. A further orientation adjustment was required the next day and DSN tracking resumed on May 21, 2003. The first JPL solution included six passes of degraded range data and 7 DDOR measurements, all on the East-West baseline. These data were used in support of DV16, performed in two parts on May 30, 2003, just nine days after reestablishing contact with the spacecraft. This 8.1 m/s maneuver corrected the altitude of closest approach by 4800 km and adjusted the arrival time earlier by an additional 10 minutes (Figure 6).

The final trim prior to Earth swingby 2 was DV17, a 0.4 m/s maneuver executed on June 16, 2006 (see Figure 7). The JPL solution for the maneuver was a 23-day data arc including 19 passes of degraded DSN range data plus 16 DDOR measurements on the East-West baseline (Figures 8a and 8b). This maneuver corrected an error of 100 km.

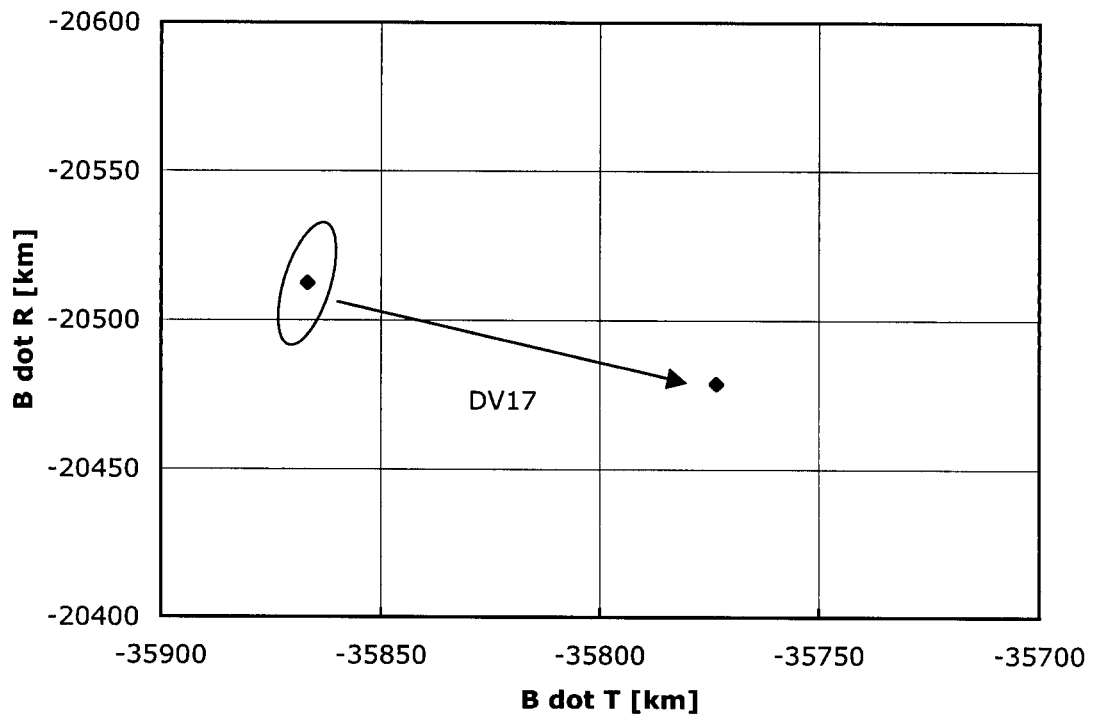
Earth swingby 2 occurred at June 19, 2003. This event was not observed because the spacecraft orientation precluded the receipt of navigation data for four days centered on periapsis. Orbit determination using DDOR and degraded range data from before and after the swingby found that closest approach occurred on June 19, 2002 14:44:33 UTC at an altitude of 11022.7 km. The B-vector error was 7.1 km.

Another nine days of range data were acquired including two DDOR measurements (one on each baseline) to verify the outgoing asymptote. It was determined that a post flyby correction maneuver was not needed. Nozomi would arrive at Mars on December 13, 2003 18:00 UTC at an altitude of 300 km.

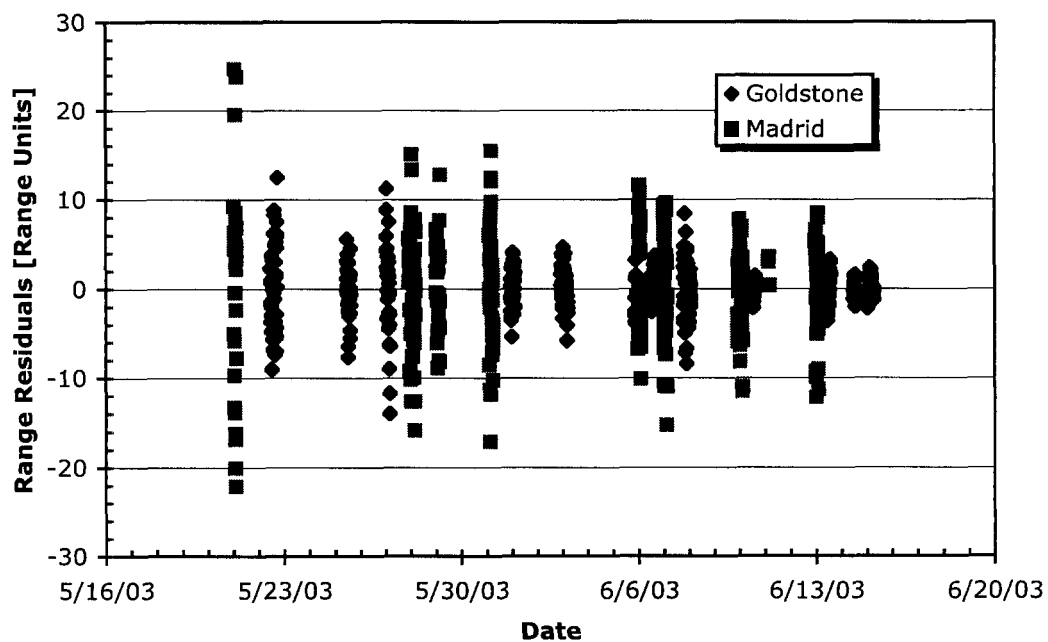
A list of Earth Swingby 2 phase events is presented as a list in Appendix 3 and in graphical format in Appendix 4.



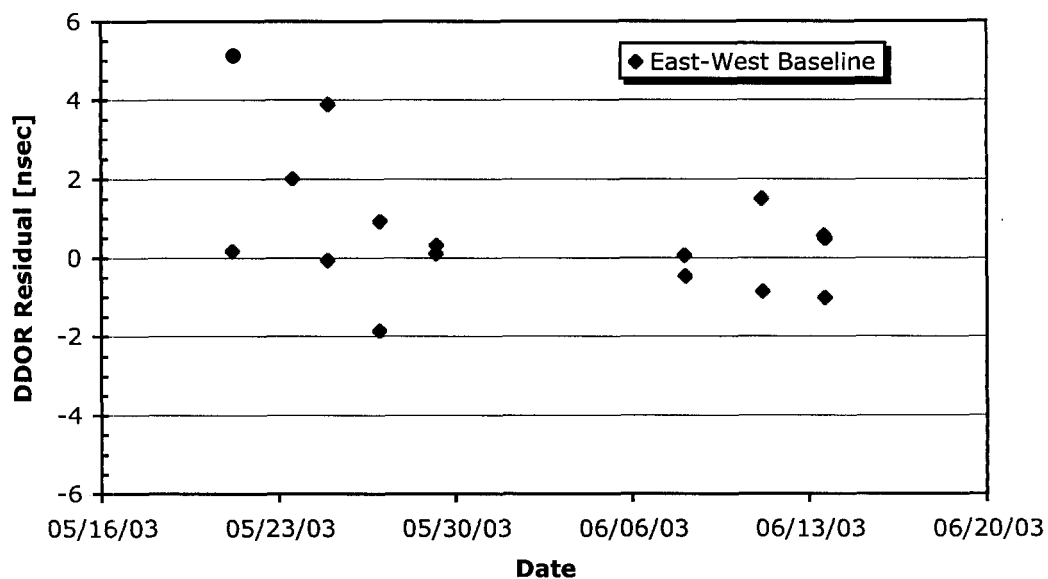
**Figure 6 Earth Swingby 2 B-Plane**



**Figure 7 Earth Swingby 2 Final B-Plane**



**Figure 8a Earth Swingby 2 Doppler Residuals**



**Figure 8b Earth Swingby 2 DDOR Residuals**

## **AFTERMATH**

Entering Mars orbit and acquiring science data would require subsystems attached to the shorted bus. ISAS resumed their effort to repair the short circuit shortly after Earth swingby 2. Once again, the downlink was disabled during this operation. ISAS would periodically suspend the repair operation to see if the repair had been achieved. But no indication of success was ever received. The Japanese also found that the downlink could not be restored.

ISAS was forced to abandon its attempt to repair the shorted bus and restore the downlink on December 10, 2003. ISAS then issued commands in the blind to perform a 2 m/s maneuver to raise the altitude of closest approach to Mars by 500 km. This maneuver was performed to reduce the probability that spacecraft might impact the planet to less than 1%. Nozomi flew past Mars on December 13, 2003 and entered solar orbit.

## **ACKNOWLEDGEMENTS**

Nozomi's delivery to Mars is a direct result of the contributions of many people. The authors would like to express their appreciation and deep respect for the members of the ISAS navigation team. Their technical excellence, sincere cooperation and heroic efforts were directly responsible for the success of the navigation task. The ISAS team members include Jun'ichiro Kawaguchi, Takaji Kato, Tsutomu "Ben" Ichikawa, Makoto Yoshikawa, Shiro Ishibashi and Takafumi Ohnishi. Thanks also go to the DDOR team for implementing the "poor mans" DDOR capability, operational interfaces and cogent advice. They include Jim Border, Jean Patteson, Walid Majid and Peter Kroger. A large number of individuals, usually unrecognized, deserve mention for their years of consistent support. They include Albert Chang, Jordan Ellis, Tim McElrath, Teresa Thomas, Connie Dang, Belinda Arroyo, Sohpie No, Eve Kenney, Melvin McBride, Robert Brodtkin, Ruben Espinueva, Ida Millner and John Covate.

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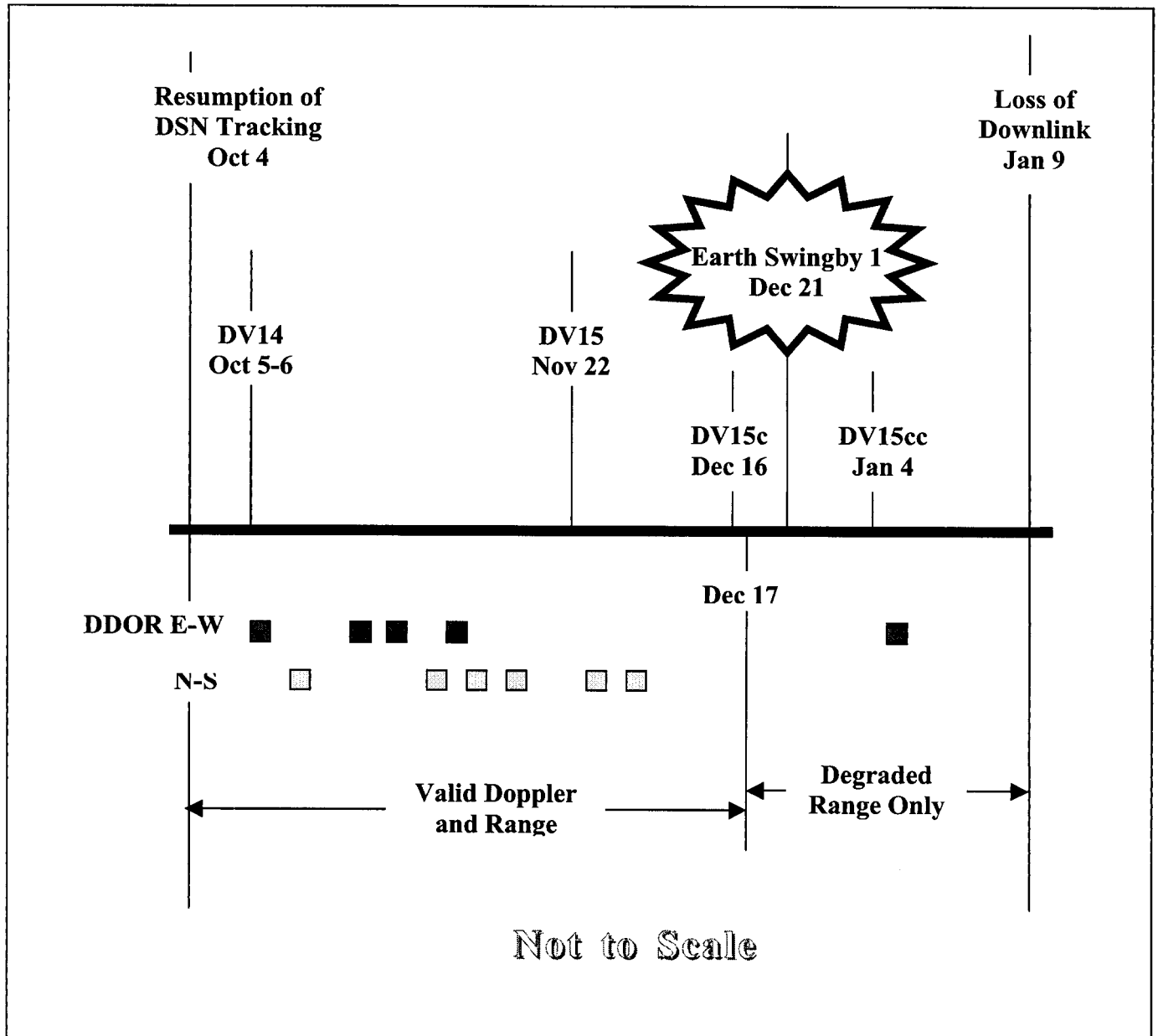
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## APPENDIX

### Appendix 1

Date (UTC)	Event	Note
05-OCT-2002 17:00:00	dv14 part 1	9.21 m/s
05-OCT-2002 20:15:00	dv14 part 2	4.03 m/s
06-OCT-2002 17:00:00	dv14 part 3	14.43 m/s
15-OCT-2002 16:19:38	re-orientation	
22-NOV-2002 11:14:18	dv15 part 1	3.0328 m/sec
22-NOV-2002 11:32:43	dv15 part 2	1.8176 m/sec
13-DEC-2002 12:00:00	re-orientation	
16-DEC-2002 11:00:00	dv15c	1.3 m/sec
17-DEC-2002 10:00:00	85° off Earth point	
18-DEC-2002 10:00:00	97° off Earth point	
19-DEC-2002 10:00:00	105° off Earth point	
21-DEC-2002 07:36:59	Earth Swingby 1	Alt = 29509.5 km
04-JAN-2003 02:55:00	dv15cc part 1	3.1 m/sec
04-JAN-2003 05:11:00	dv15cc part	0.2 m/s
28-JAN-2003 12:00:00	57° off Earth point	

## Appendix 2



### Appendix 3

Date (UTC)	Event	Note
30-MAY-2003 02:09:30	dv16	8.086 m/s
16-JUN-2003 20:00:00	dv17	0.372 m/s
19-JUN-2003 14:44:33	Earth swingby 2	Alt = 11027.7 km

### Appendix 4

